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PLANETARY MISSIONS: A LOOK AT ENTRY VEHICLE PROBLEMS

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Introduction

The exploration of Earth's neighbor planets Mars and Venus presents one of the most challenging and at the same time one of the most rewarding objectives of our space program. The planets have evolved differently as a result of their relative positions in the solar system and it is of extreme interest, scientifically, to determine how the resulting environment of Mars and Venus differs from that of Earth. The character of the planetary atmospheres, the details of surface features and the possible existence of extraterrestrial life are of particular interest.

Such detailed information can be obtained only by placing measuring instruments into the atmosphere and on the surface of the planet, however, and it is clear that spacecraft designed for atmospheric entry and surface landing will play an important role in the planetary exploration program. Furthermore in the event of manned missions to Mars and Venus, the requirements for return to Earth again emphasize the atmospheric entry aspects of the exploration program.

In the last decade a large research and development effort has been applied to the design of entry vehicles, both unmanned and manned,

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for use in the Earth's atmosphere and much of the technology produced by this effort can be applied directly to the planetary exploration program. There are some significant differences, however, between the needs of past and current entry vehicles and those expected of vehicles required for the future planetary program, and these arise primarily because the characteristics of the planetary atmospheres are different and because in some cases the entry velocities will be greater than those previously experienced.

We can expect that entry vehicles to be used in planetary missions will be tailored to meet these new conditions, and the unmanned Mars entry vehicle, for example, may bear little resemblance to the ICBM nose-cone and will therefore require a somewhat different design approach.

In this article an attempt is made to review some of the research and development problems that arise in the design of both unmanned and manned planetary entry vehicles. Greatest attention is given to unmanned Mars vehicles in view of the imminent needs of the planetary program, and less detailed consideration is made of manned vehicles since our design concepts will undoubtedly change as more information on the planetary environment becomes available and as the technology advances during the next ten to twenty years.

Unmanned Missions

The exploration of a planet whose atmosphere is not well defined poses the problem of deciding what the first vehicle should be: an

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entry vehicle is required to place any measuring equipment into the lower atmosphere and on the surface, but at the same time the atmospheric measurements are to some extent required for proper design of the entry vehicle. The first step must clearly be one that we can take with some confidence of success, and should yield information of scientific value (for example, measurements on the atmospheric composition of the lower atmosphere that can shed light on the possible existence of extraterrestrial life) and information that will contribute to the design of future experiments in the program. Moreover, the design concepts for the first mission vehicle should have sufficient potential to allow growth into the subsequent, more elaborate, mission vehicles required later in the program.

The design of the entry vehicle must evolve from a number of inputs. Foremost among these should be a definition of the purpose of the mission including the measurements to be made, the instruments to be used and the manner in which data communication is to take place. In this way the internal payload can be described and the required data acquisition period, and if the vehicle is to land the permissible impact decelerations, can be established. Existing information on the planetary atmosphere, (i.e., our best estimates of atmospheric pressure, scale height, composition, etc.) together with the expected entry conditions, particularly entry velocity, allows us to determine within reasonable bounds the problems associated with entry loads, entry heating, communication blackout, terminal impact, etc., and these place further constraints on the configuration. The final design will be, undoubtedly, a compromise which respects the technical problems involved and still allows a worthwhile mission to be accomplished.

As a point of departure into a discussion of entry vehicle design problems we can consider a spherical atmospheric probe such as that suggested by Sieff[(1)], and more recently elaborated by Buef[(2)], for use in the 1966 Mars mission. The purpose of this spherical probe (see fig. 1a) is to acquire information on the atmospheric properties by observing the motion of the sphere as it decelerates due to atmospheric drag. The deceleration (which is independent of the attitude for a sphere) is measured and transmitted to the parent fly-by spacecraft for relay to Earth, data communication taking place after the probe emerges from blackout over a period of about 30 seconds. The velocity during entry is found by integration of the deceleration history and the altitude by a further integration with respect to time. From this information the variation of density and pressure can be determined from the drag equation. Additionally, some information regarding the composition can be determined by observation and analysis of the spectrum of radiation emitted by the shock layer that surrounds the sphere during entry. Such an atmospheric probe appears feasible at the present time, providing it can be placed on the appropriate flight path to ensure a fairly steep entry angle, but since the atmospheric properties are not measured directly the data obtained are not very accurate: an error of $\frac{1}{2}\%$ in the accelerometer measurement for example leads to a 100% error in the value obtained for the atmospheric density.

As we look ahead to missions beyond 1966 it is clear that more extensive measurements will be required, both in the atmosphere and on the planet surface, with correspondingly longer periods for data communication. At the 1969 launch opportunity using the Atlas Centaur Launch Vehicle it should be possible to use a Probe/Lander (Fig. 1b) entry vehicle to determine, in detail, conditions in the lower atmosphere of Mars; additionally preliminary information on surface conditions would be obtained over the time period that the lander remains in view of the fly-by spacecraft. Such a Probe/Lander would be a very lightly loaded high-drag vehicle and would require a terminal parachute in order to decelerate sufficiently in the tenuous Mars atmosphere. The information gained from such a mission would be used in the formulation of experiments, and for the design of sensors and other instruments to be used for future missions, possibly an Automated Biological Laboratory or an Automated Weather Station. These advanced Lander vehicles would soft-land with a retro-rocket system, operate for many months, possibly a year, and could communicate directly with Earth or use a Planetary Orbiter as a relay link (see Fig. 1c).

In view of the simplicity of the sphere it is interesting to determine whether it has general utility as an entry vehicle for use in Mars and Venus missions. The history of this sphere during vertical entry into a planetary atmosphere is depicted in figure 2 where the velocity is plotted against an atmospheric drag parameter (essentially a ballistic parameter introduced by Allen and Eggers[3]. This drag parameter can be interpreted in the following way. During the time in which the sphere decelerates from the entry velocity V_e to a velocity V it encounters a mass m_{atm} of the atmosphere. The drag parameter in figure 2 is simply the ratio $\frac{m_{atm}}{m}$ and can be written

$$\frac{m_{atm}}{m} = \frac{\int \rho C_D A dh}{m} = \frac{p}{mg/C_D A}$$

where ρ, p and h are respectively the density, pressure and altitude in the atmosphere, g is the planet gravitational acceleration, $C_D A$ is the effective drag area of the sphere.

During atmospheric entry the vehicle undergoes aerodynamic heating, sustains aerodynamic loads (these are most severe at $\frac{m_{atm}}{m}$ equal to $\frac{1}{3}$ and 1 respectively) and continues to decelerate until it passes through sonic conditions and finally, if there is sufficient atmosphere, reaches a terminal condition $\left(\frac{m_{atm}}{m}\right.$ greater than about 10 typically$\left.\right)$ in which the drag force is balanced by the gravitational force.

It is clear from figure 2 that the vehicle velocity decreases rapidly as the ratio $\frac{m_{atm}}{m}$ or $\frac{p}{mg/C_D A}$ becomes larger, and it follows that the value of $\frac{m}{C_D A}$ required to decelerate the vehicle to terminal conditions varies directly as the atmospheric pressure p. Now for

Mars, the atmospheric pressure near the surface is presently thought to be as low as 10 mbars (compared with approximately 1,000 mbars for Earth) and as a consequence terminal conditions are achieved only if $\frac{m}{C_D A}$ is less than about $.2 \frac{\text{slug}}{\text{sqft}}$. For Venus, however, the surface pressure is thought to be between 1 and 3 atmospheres and terminal conditions are achieved even if $\frac{m}{C_D A}$ is as large as $20 \frac{\text{slugs}}{\text{sq ft}}$.

The design parameters which characterize the spherical probe are the ballistic parameter $\frac{m}{C_D A}$ and the diameter D and it is convenient to discuss the sphere in terms of these quantities. The constraints which determine the permissible range of these parameters depend on the entry velocity and entry angle and on the atmospheric properties. The expected entry velocity is dictated by the interplanetary trajectory and is known fairly accurately and for the present it is assumed that the entry path is vertical. If the atmospheric properties are assumed known (as they must be in order to arrive at a design) then all the factors which determine the constraints can be expressed in terms of $\frac{m}{C_D A}$ and D.

In view of the significant differences between Mars and Venus with respect to both the expected entry velocity and the atmospheric pressure the problems are discussed separately for the two planets.

Taking first the case of Mars, the most significant constraint results from the low atmospheric pressure at the surface. A sphere of conventional mass to area ratio, $\frac{m}{C_D A} = \frac{1 \text{ slug}}{\text{sq ft}}$ entering the Mars atmosphere (having a surface pressure of 11 mbars) at 25,000 ft/sec would spend only a few seconds in the atmosphere and impact the surface at about 10,000 ft/sec. Furthermore, under these conditions it would suffer communication blackout throughout most of its flight and would impact before emerging from blackout. Clearly if the probe is to have sufficient time to gather and communicate data it must have a value of $\frac{m}{C_DA}$ significantly less than $\frac{1 \text{ slug}}{\text{sq ft}}$; as the value of $\frac{m}{C_DA}$ is reduced the available data communication time is increased and for $\frac{m}{C_DA} = .25 \frac{\text{slug}}{\text{sqft}}$ the period between emergence from blackout and impact is about 30 seconds. It seems unlikely that a significant amount of information can be obtained in less time than this.

In order to obtain a longer data collection period the value of $\frac{m}{C_DA}$ must be further reduced and for a given drag area, C_DA, this implies a reduction in payload mass. Here, however, it should be remembered that the sphere must withstand both aerodynamic loads and aerodynamic heating and a limit is soon approached in which the entire mass is assigned to the load carrying structure and to thermal protection leaving no mass available for internal payload.

The aerodynamic loads for vertical entry into the Martian atmosphere, at a typical entry velocity of 25,000 ft/sec, are expected to be about 200 Earth g's and the convective heating approximately the same as that for Earth entry. Additional heating associated with radiation from the gas layer surrounding the vehicle depends on the atmospheric composition, and for N_2-CO_2 mixtures (likely candidates for the atmospheres of Mars and Venus) it has been shown that radiative heating is significant at entry speeds as low as 20,000 ft/sec, whereas for Earth entry this form of heating is not significant even at 30,000 ft/sec. For spheres of large diameter the structural weight increases more rapidly than the surface area (due to the increased thickness of the structural shell required to maintain stiffness and so avoid compressive buckling) and increased radiative heating places greater demands on the thermal protection system (radiative heating per unit area increases directly as the diameter).

For spheres of small diameter the structural weight becomes less significant but the convective heating (which varies as $D^{-1/2}$) requires a greater thickness of ablation material and the low $\frac{m}{C_D A}$ sphere can do little more than provide itself with sufficient protection to survive entry heating. Figure 3, which is plot of $\frac{m}{C_D A}$ vs D on logarithmic scales, indicates how the designer is literally boxed in by these constraints when he attempts to design a spherical entry vehicle. The lower boundary corresponds to an internal payload of $\frac{1}{2}$ slug representing about the minimum mass required for a power supply and communication system. Along the upper boundary of the design box, where $\frac{m}{C_D A} = .25$, the minimum diameter is 2 ft. and the maximum useful diameter is 8 ft. corresponding to a payload of 3.5 slugs, the maximum possible for any sphere within these constraints: spheres of larger diameter would be structurally too heavy to carry 3.5 slugs and stay within the $\frac{m}{C_D A}$

constraint. (The Centaur shroud diameter is also shown in fig. 3 but does not provide a significant constraint since the larger spheres would actually correspond to reduced payloads.)

The Mars spherical probe appears to be feasible if the internal payload has sufficiently small mass and if the data collection period is of the order of 30 secs but it offers little promise of being useful for the larger payloads envisioned in future missions.

Considering now the Probe/Lander system depicted in figure 1b, the primary constraint arises from the need to achieve a sufficiently low velocity to allow deployment of a parachute; if a conventional parachute is employed the vehicle must decelerate to subsonic speeds before reaching the surface and this requires that $\frac{m}{C_D A}$ be less than $\frac{1}{4}$ slugs/sq ft. If in addition the vehicle is to collect appreciable atmospheric information before landing, it is desirable to reduce $\frac{m}{C_D A}$ even further (to a value of about .15 slugs/sq ft). The limitations of the sphere as a low $\frac{m}{C_D A}$ entry vehicle lead us to consider other shapes which for one reason or another are more efficient. For a given area A the allowable mass increases with the drag coefficient, and the mass of the internal payload increases further when the structural weight and the thermal protection weight (both of which vary as the surface area) are minimized.

Ideally, then, the vehicle should have high drag and small exposed surface area and still enclose sufficient volume to contain the payload.

A flat disc has the highest drag ($C_D \approx 1.8$) for given surface area but has no volume, whereas the sphere contains the maximum volume for given surface area but has insufficient drag. Evidently the ideal vehicle is a combination of flat disc with a sphere of sufficient volume to contain the internal payload. In general, because the required mass to area ratio must be small the sphere needed to contain the payload has relatively small diameter (compared with that of the disc). A combination of a disc and such a sphere, suitably rounded off to make it aerodynamically respectable, leads to either an Apollo-shaped vehicle or to a shallow blunted cone as seen in figure 4. Such shapes have drag coefficients of the order of 1.5 and have low exposed surface areas, S, leading to low thermal protection requirements (generally speaking, a small value of the parameter $\frac{S}{C_D A}$ is desirable and the shallow blunted cone has a value .8 compared with 2 for a sphere). The blunted cone tends to have better aerodynamic stability than the Apollo shape, especially if the internal payload mass can be placed at the bottom of the cone, and for this reason the trend for future unmanned Mars vehicles is likely to be in this direction.

Having reduced the total surface area in this way further increases in payload can be realized only by reducing the structural and thermal protection weights per unit area. The thermal protection material must itself satisfy a number of requirements, for example, it must be capable of withstanding the long "cold-soak" experienced during the spaceflight, it must have high thermal performance during entry and if possible should be such as to allow communication after entry. Such materials are presently available (compounded elastromeric materials) but the choice is extremely limited and there is little possibility of reducing the thermal protection weight by any appreciable amount.

The structural weight, especially for vehicles of large diameter, becomes the most significant part of the total vehicle weight for any structure under compressive loads. This is clear from figure 3 for a sphere (since increasing the diameter merely increases the structural weight and actually reduces internal payload) and this is true to some extent for all vehicles subject to compressive buckling. Under tensile loads however the structural weight can be reduced significantly and a "tension structure" for future unmanned Mars vehicles has recently been developed at Langley Research Center as reported by Anderson[4]. The structure consists of a circular compression ring and a spherical cap joined by a shell (see fig. 5a) whose shape is such as to be in tension when aerodynamically loaded. The structural weight of this configuration is less than 30% of a sphere having the same base diameter and, for large vehicles especially, can lead to a major increase in payload. This concept has been tested to verify both its structural performance and its aerodynamic high-drag performance. In hypersonic flow a detached shock appears around the rear of the vehicle (as seen in fig. 5b) and gives rise to higher pressures over a substantial fraction of the total frontal area (typical drag coefficients between 1.25 and 1.5).

The internal payload mass available, comprising the payload and retardation system, when the tension structure is used, is shown in figure 6 together with the ranges of $\frac{m}{C_D A}$ in which parachute and retro-rocket systems are required in order to achieve a survivable landing. A Mars Probe/Lander such as that indicated in figure 1b for the 1969-1971 period having a diameter of about 8 ft. (within the Centaur shroud) would have a maximum internal payload between 3 slugs and 12 slugs. For the minimum vehicle (3 slugs, representing minimum instrumentation, communication system, etc.) approximately 10 minutes of data collection time within the atmosphere would be available in addition to a period of several hours on the surface. The vehicle having maximum payload (12 slugs) would have negligible data collection time within the atmosphere but would allow a heavier instrument package to be placed on the surface. In each case approximately half of the internal payload comprise instrumentation and the communication system, the remaining half being required for the parachute and landing structure. Payload weight could be increased beyond 12 slugs using a larger vehicle (up to 20 ft. in diameter, say) but such a vehicle would be extremely inefficient since most of the mass would be taken up by the structure and thermal protection system.

More advanced vehicles such as an Automated Biological Laboratory or an Automated Weather Station may be appreciably heavier than the earlier Probe/Lander system and tend to become extremely large in size if aerodynamic braking is the primary means of retardation. The situation can be improved to some extent if a shallow entry path can be chosen (entry at an angle of 30° allows the $\frac{m}{C_D A}$ to be twice that for vertical entry) but the use of aerodynamic retardation after direct entry from the interplanetary trajectory still places a severe constraint on $\frac{m}{C_D A}$ (and therefore on payload mass) and it is clearly appropriate to use some propulsive retardation, especially if Saturn class launch vehicles, with their increased capability are used.

There remains the question whether to use propulsive braking before or after atmospheric entry. If direct entry is made at an entry angle of 30°, even for $\frac{m}{C_D A} = 2$, the vehicle will approach the surface at 10,000 ft/sec and if this is reduced to zero by propulsive retardation the terminal vehicle weight is reduced to roughly a quarter of the entry weight. The alternative is to apply the 10,000 ft/sec propulsive velocity increment prior to entry thus allowing the vehicle to establish a near circular orbit about Mars. Subsequent entry by orbital decay allows appreciable atmospheric retardation even for entry vehicles of conventional design: furthermore aerodynamic heating and aerodynamic loads are less severe since the entry velocity is now of the order of 12,000-15,000 ft/sec rather than 25,000 ft/sec. If it is verified that the atmosphere is as tenuous as is presently suggested this latter course is clearly preferable. Terminal descent to the surface would be accomplished with a secondary retro-rocket system.

Turning briefly now to the use of the sphere as a Venus atmospheric probe, the ballistic parameter $\frac{m}{C_D A}$ determines the altitude at which the sphere emerges from blackout; thus if it is required to determine properties above the cloud layer in the Venusian atmosphere the $\frac{m}{C_D A}$ must not be too large. On the other hand, in view of the relatively large value of the atmospheric pressure at the surface, a sphere of $\frac{m}{C_D A} = 1 \frac{\text{slug}}{\text{sqft}}$ would have an appreciable data collection period in the lower atmosphere.

The primary constraint on vehicle design for a Venus probe results from the high entry velocity (in excess of 40,000 ft/sec) and the corresponding high convective and radiative heating rates sustained during entry. Both convective and radiative heating increase with the ballistic parameter $\frac{m}{C_D A}$. Figure 7 is a plot of $\frac{m}{C_D A}$ against D and illustrates the several constraints which determine the permissible design range. A minimum internal payload curve ($\frac{1}{2}$ slug) shows that the sphere must have sufficiently high $\frac{m}{C_D A}$ to provide structural and thermal protection mass, that the diameter must be sufficiently large to avoid excessive convective heating but sufficiently small to avoid excessive radiative heating and excessive structural weight. Furthermore it cannot be too dense or problems arise with the packaging of scientific equipment.

Increases in payload mass can be realized only by increasing $\frac{m}{C_D A}$ but here a limit is set by the fact that the vehicle Reynolds' number increases with $\frac{m}{C_D A}$ and turbulent heating results, with a corresponding increase in thermal protection requirements (the boundary indicated in figure 7 is for a Reynolds number R_D, of 10 million). Once more the designer is boxed in by constraints in such a way that the maximum payload that can be carried by a spherical probe is about 2 slugs.

These constraints are more severe than those for the Mars probe and the means of enlarging the design range are not completely clear: radiative heating can be reduced appreciably by using a cone-shaped entry vehicle as shown by Allen[5] but even when radiative heating is discounted, turbulent heating and large structural weight preclude the use of larger or heavier vehicles. The alternative is to reduce $\frac{m}{C_D A}$ and use a lighter structure, so again the low $\frac{m}{C_D A}$ tension structure, now having small nose radius, offers a possible solution.

The successful development of entry vehicles for the Planetary Exploration Program depends to a large extent on the strength of the supporting research and technology program. Instrumentation for the first missions has, to a large degree, already been developed and in some cases flight tested; the question of the performance of these instruments after sterilization, at present a requirement for all planetary vehicles, is still unanswered however. Our understanding of the problems of communication from an entry vehicle is such that data transmission before and after 'blackout' can be made with little difficulty. In recent years considerable experience has been gained in this area in connection with Earth entry flight programs, for example the RAM Program and Project Fire, and the techniques developed there can be used with little modification. The aerodynamics of the spherical probe are well known, of course, but for more complex shapes further research is needed on the aerodynamics, (drag, stability, convective and radiative heating through a large range of angle-of-attack), on the stability of lightweight structures and on aeroelastic effects which may be associated with the vehicle oscillation during atmospheric entry.

The problem of thermal protection for Mars entry vehicles is similar to that for Earth entry from a circular orbit and is not of major concern. For Venus probes, however, entry heating, particularly radiative heating, is much more severe and the thermal performance of the ablation shield is a critical factor in vehicle design. The mechanism of ablation under conditions of high radiative heating rates and high aerodynamic shear stress is not completely understood; furthermore, present test facilities cannot produce the appropriate entry environment and the lack of experimental data has prevented much progress in this area. The material needed here must sustain high surface temperatures to re-radiate a substantial amount of heat, it must have a large heat of sublimation and must resist mechanical shear. Of the many materials considered, phenolic nylon and phenolic graphite appear as likely candidates and are presently included in a program of ground and flight tests to be carried out at Langley Research Center.

The development phase for unmanned entry vehicles requires an extensive test program, especially in view of the requirement for sterilization and the uncertainty that this introduces to the reliability of components and subsystems. For the Mars entry vehicles much of the system development can be carried out in ground facilities but a final test of the complete prototype vehicle in the Earth's atmosphere would qualify the system for mission use. Such a test, with an appropriately chosen entry angle, would simulate aerodynamic loads, entry dynamics, convective heating and blackout, and would verify the performance, after sterilization, of the integrated system including the heat shield and structure, instrumentation, communication system and parachute.

Manned Missions

Although manned missions to the planets are much further in the future than the unmanned missions they have already become the subject of considerable analysis and discussion. Because of the current view that Mars offers a more hospitable climate than Venus, most studies of manned missions have considered landing on Mars and returning to Earth after a short period of exploration. Such studies, even one or two decades before the mission appears likely, are valuable inasmuch as they allow an assessment of program needs particularly with respect to new launch vehicles and advanced propulsion. Detailed studies of subsystems, even of a major subsystem such as the Earth Entry Vehicle, can represent only the present state of the art, however, and with the rapid growth in technology that undoubtedly will occur during the next twenty years, the mission article of 1980 may bear little relation to the concept of 1964.

With these reservations in mind the following discussion will attempt to summarize the present position with respect to Manned Entry Vehicles and to make tentative suggestions as to the possible directions that our research and development effort will take in the entry vehicle technology area.

The use of aerodynamic braking has long been recognized as a desirable alternative to propulsive braking as a means of reducing overall mission weight or alternatively of reducing overall mission duration. More recently studies have suggested that entry velocities into the Earth's atmosphere (upon mission return from Mars) which tend to be in the range 50,000 to 70,000 ft/sec for missions of short duration can be reduced to the range 43,000 to 49,000 ft/sec by the use of what has been termed the Venus Swingby Mode - this mode, reported recently by Syvertson and Dennis[6], essentially reshapes the return trajectory by passing through the Venus gravitational field to one more aligned with the Earth's path around the Sun. The variation within this range results from the changing relative positions of Earth, Mars, and Venus during a 15 year cycle. Entry velocities into the Martian atmosphere lie in the range from 20,000 ft/sec to

30,000 ft/sec for direct entry and braking into orbit and in the range 12,000 ft/sec to 16,000 ft/sec for descent from a captured elliptic orbit about Mars.

Aerodynamic braking, while extremely useful, also carries its own penalties of course and these are bestowed upon the entry vehicle. If atmospheric braking is used as a means of establishing a Mars capture orbit, the "entry vehicle" is in fact the whole mission vehicle including a Mars Excursion Module, the Spaceflight Living Module, the Return Propulsion Module and the Earth Entry Vehicle and this whole system must be designed to withstand entry loads and entry heating.

The vehicle design involves a compromise between entry guidance and control problems, aggravated here by possible fuel-slosh effects, and the aerodynamic problems associated with the provision of sufficient lift to ensure a wide enough entry corridor. For entry at 25,000 ft/sec and an allowable undershoot deceleration of 6 Earth g's, a lift-to-drag ratio, $\frac{L}{D} = .3$ is required to allow a 1^{o} range of entry angle. This varies approximately linearly so that $\frac{L}{D} = 1$ is required for a 3^{o} range in entry angle. Clearly the additional structure and thermal protection weight necessary to convert the space vehicle to a lifting entry vehicle will be more than offset by the saving in fuel weight required by retrobraking, but from an operational point of view it may not be desirable to commit the entire vehicle to the severe environment of atmospheric entry half way through the mission. Once the Martian orbit has been established entry of an Excursion Module does not involve either extreme heating or extreme loads and the design of this vehicle is well within the present state of the art.

Earth entry, at the termination of the mission, is quite a different problem however and represents a major step beyond Apollo: even entry velocities of 45,000 ft/sec are expected to produce substantially greater heating rates due primarily to gas-cap radiation and this effect together with the greater need for aerodynamic lift will be reflected in the entry vehicle configuration.

Radiative heating can be alleviated, at the expense of moderate increases in convective heating, by providing the vehicle with a forebody that has a small nose radius and is swept back to ensure that the component of velocity normal to the shock is appreciably reduced. The width of the entry corridor between the 12g (undershoot) and the atmospheric capture (overshoot) boundaries depends primarily on the entry velocity and the vehicle $\frac{L}{D}$. Acceptable entry corridors (between 12 and 25 miles in width) can be achieved using fairly simple entry procedures even at speeds approaching 60,000 ft/sec providing the vehicle has $\frac{L}{D}$ near unity, as seen in figure 8. Increases in corridor width can be realized by the use of lift modulation as suggested by Love[7]. This requirement for $\frac{L}{D}$ approaching 1 also forces the vehicle design away from the very blunt forebody that characterizes Mercury and Apollo and toward the swept-back lifting body: fortunately this trend is compatible with that required to reduce radiative heating.

Present concepts of the Earth entry vehicle stem from modifications either to the Apollo or to the lifting $(\frac{L}{D} \sim 1)$ entry vehicle. Figure 9a shows an Apollo of elliptical cross-section matched to a conical forebody which provides lift and reduces radiative heating, whereas figure 9b shows a lifting body having aerodynamic control surfaces. Each of these configurations, and indeed any other that could be suggested at the present time, would require substantial research and development for use at entry speeds of 45,000 ft/sec or more (although it is probably true to say that they are within the present state of the art for entry up to 30,000 ft/sec).

Both the total heat input and the maximum heat transfer rates experienced during entry are so great that an ablation shield is required for all parts of the vehicle that 'see' the flow and over the nose and leading surfaces appreciable ablation would take place if any materials currently available were to be used - estimates suggest that approximately 20% of the total vehicle mass would be lost by ablation at an entry speed of 50,000 ft/sec. Presumably jet reaction controls could be used to avoid the problems that would arise with exposed aerodynamic control surfaces but even with this simplification the changes in surface shape involved could easily affect the aerodynamic characteristics of the vehicle at a critical period of the entry near peak deceleration.

A better definition of the entry heating environment, at speeds up to 50,000 ft/sec and the performance of materials in this environment, is particularly important before future mission vehicles can be designed with confidence. Questions regarding the magnitude of radiative and convective heating at high enthalpy levels, the maintenance of laminar flow over ablating surfaces, and additional radiation from ablation products can have an important effect on vehicle design and although some of the answers will be obtained in ground facilities it will be necessary to pursue flight test programs.

The recent successful flight of the Project Fire entry vehicle is an indication of the trend in this direction: in this flight detailed measurements were made of the spectral distribution of radiation from the gas cap, the total radiative heating and the convective heating experienced during entry at 37,800 ft/sec. The vehicle weighed 185 lbs. and required an Atlas launch vehicle to provide the desired high velocity at entry.

But even in flight the proper simulation of the entry environment is by no means straightforward. Manned entry is characterized by shallow angles and long duration heating (resulting from the requirement of low deceleration) and this type of entry is difficult to achieve with a ballistic vehicle. The use of lifting vehicles in such a test program is desirable but at the same time represents additional complications from the point of view of guidance and control. The Asset program has taken the first steps in this direction but the extension of this technique from speeds of 16,000 ft/sec to speeds in excess of 40,000 ft/sec would involve design problems approaching those of the mission vehicle itself.

An alternative and simpler method that is worth consideration is to use a spin-stabilized ballistic vehicle which enters at a shallow angle, passes through the atmosphere, (see fig. 10) and exits into a close earth orbit. Data would be recovered by telemetry during exit and prior to the second entry after completion of one orbit.

It is evident that future manned planetary missions will require entry vehicles that are entirely different from the existing pattern set by Mercury, Gemini and Apollo, whose origins lie in the ballistic missile technology. The future manned entry vehicles are more likely to evolve from the swept back lifing-body concepts that have recognized from the very beginning the particular needs, and the particular advantages, of man in the system. The technology for the lifting-body class of vehicles can already support flight articles at orbital speeds and with a vigorous entry technology program in the next decade it should be possible to extend their capabilities to include Earth entry from planetary missions.

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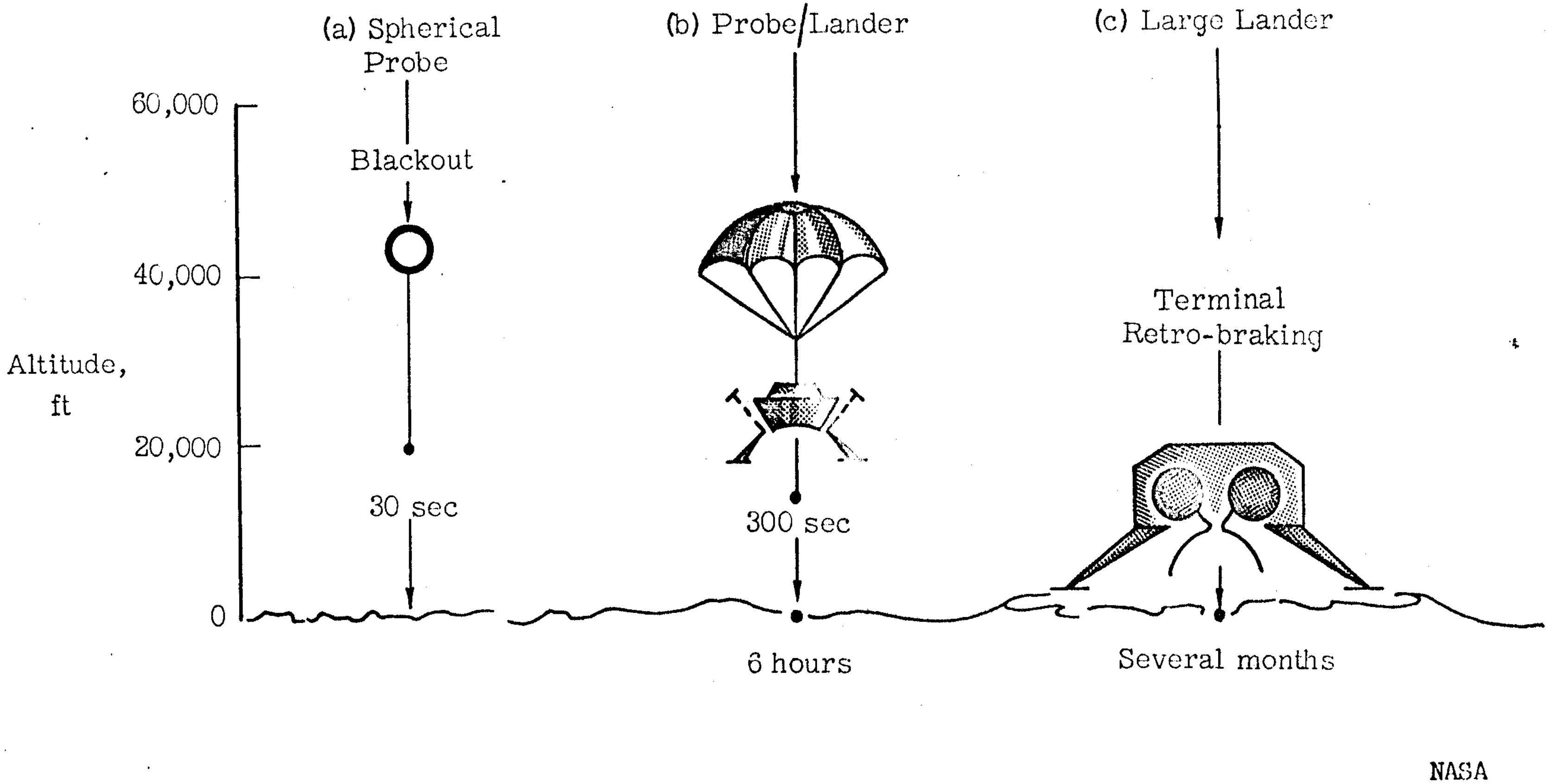

Figure 1.- Planetary probe and lander vehicles.

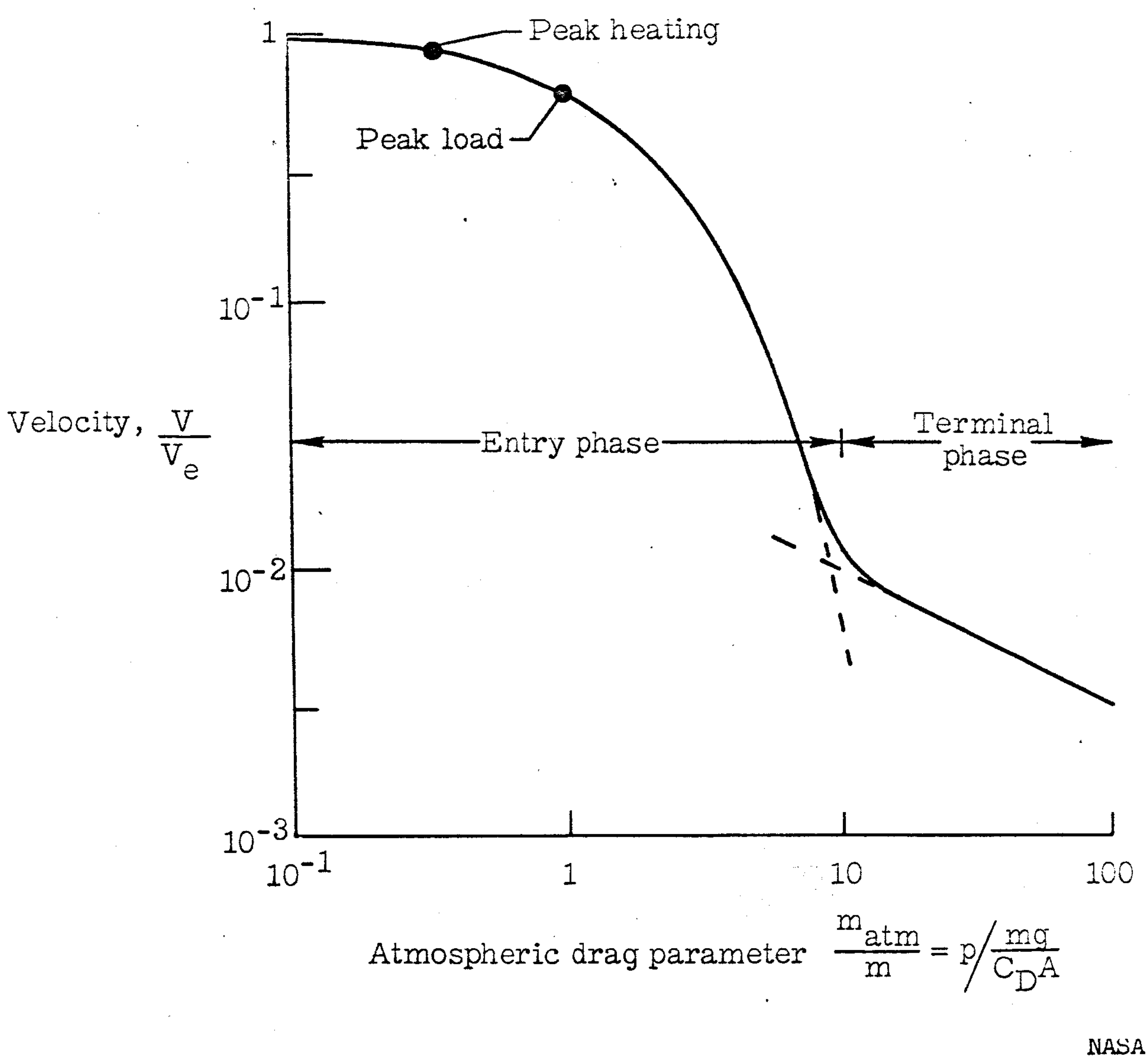

Figure 2.- Atmospheric drag parameter $\frac{m_{atm}}{m} = p\Big/\frac{mg}{C_D A}$.

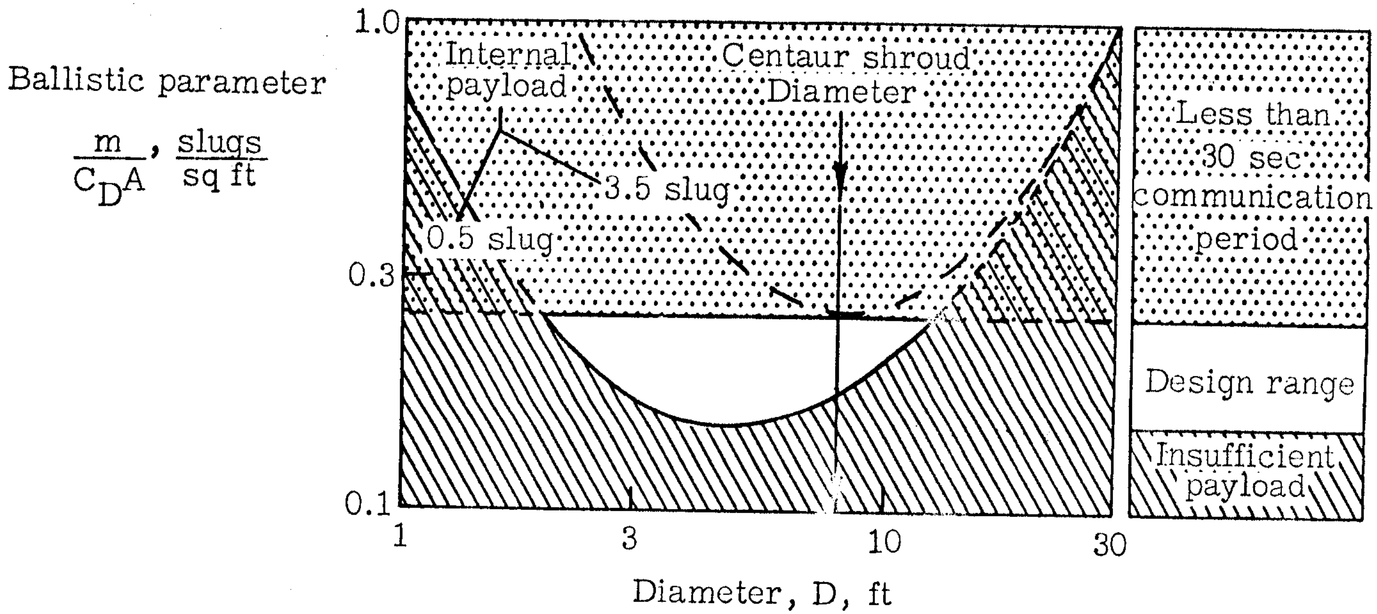

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Figure 3.- Constraints on Mars spherical probe design. (Atmospheric pressure, 11 millibars; vertical entry at 25,000 ft/sec.)

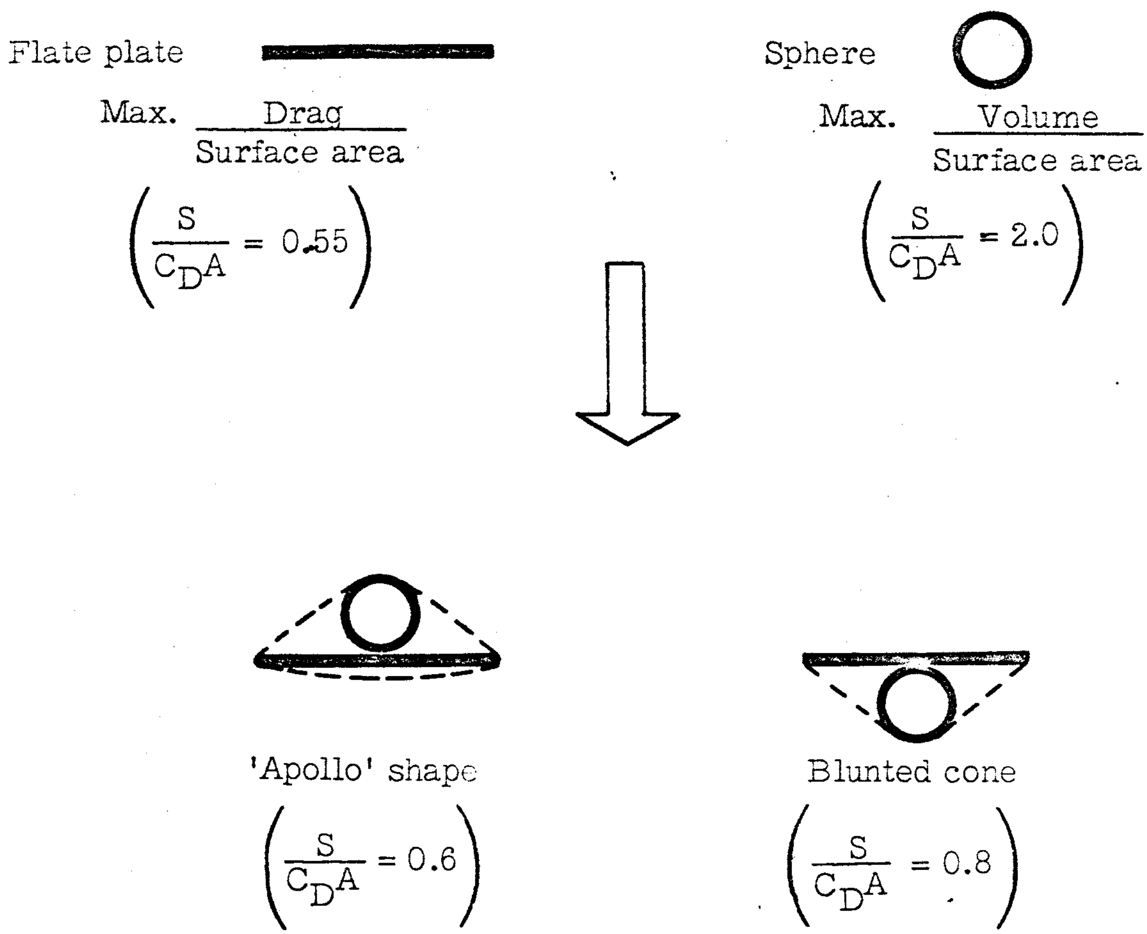

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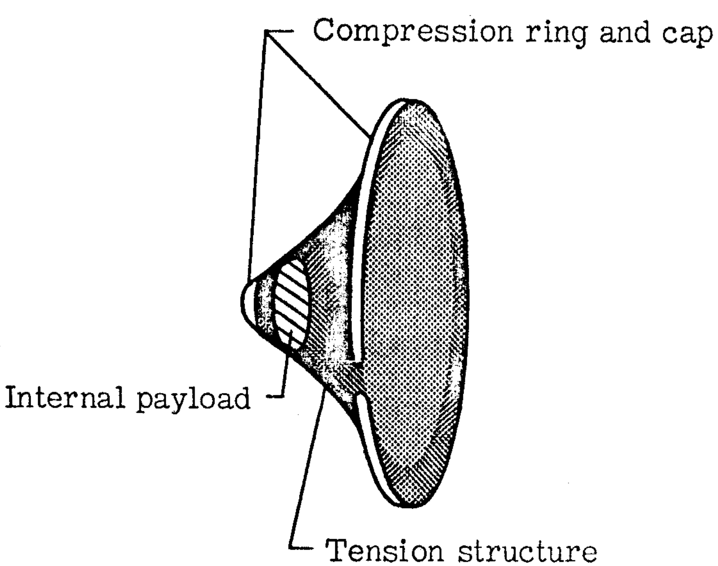
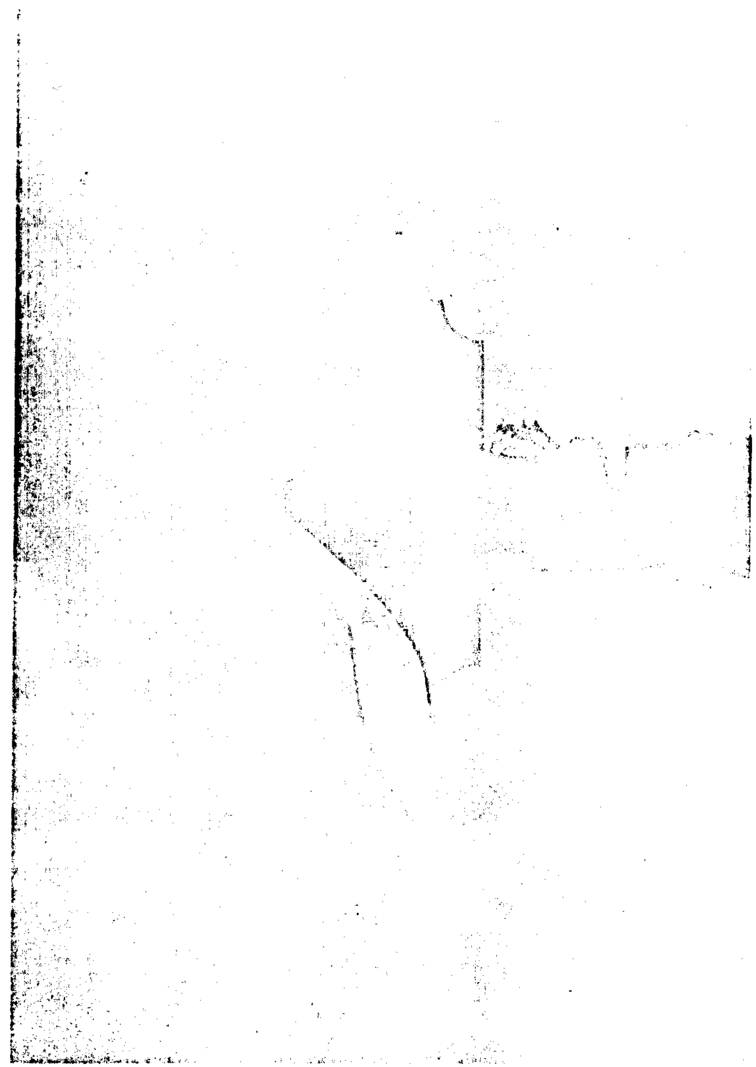
Figure 4.- High drag vehicle shapes.



(a) Tension structure.



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(b) Shock pattern at $M = 7.0$.

Figure 5.- Low m/C_DA entry vehicle concept.

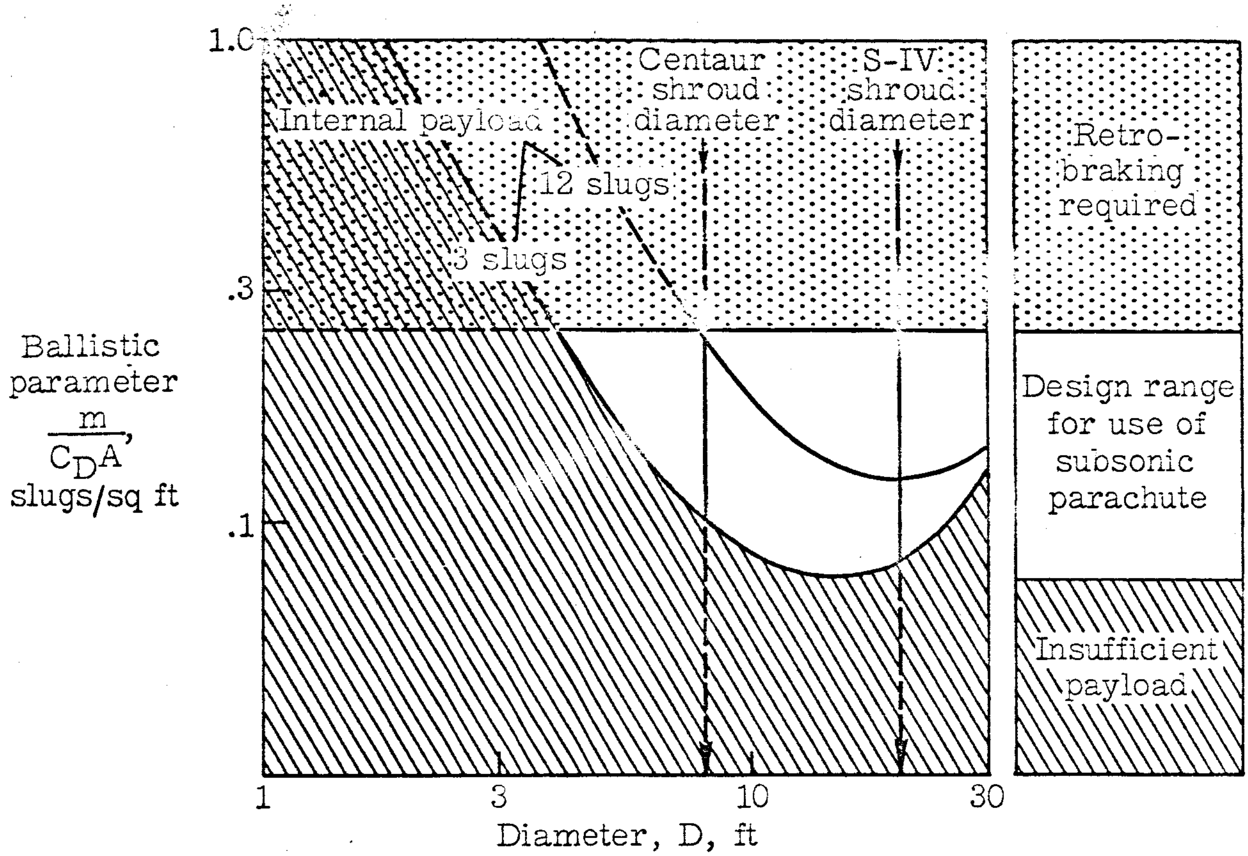

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Figure 6.- Payload capability of Mars probe lander using a tension structure. (Atmospheric pressure, 11 millibars; vertical entry at 25,000 ft/sec.)

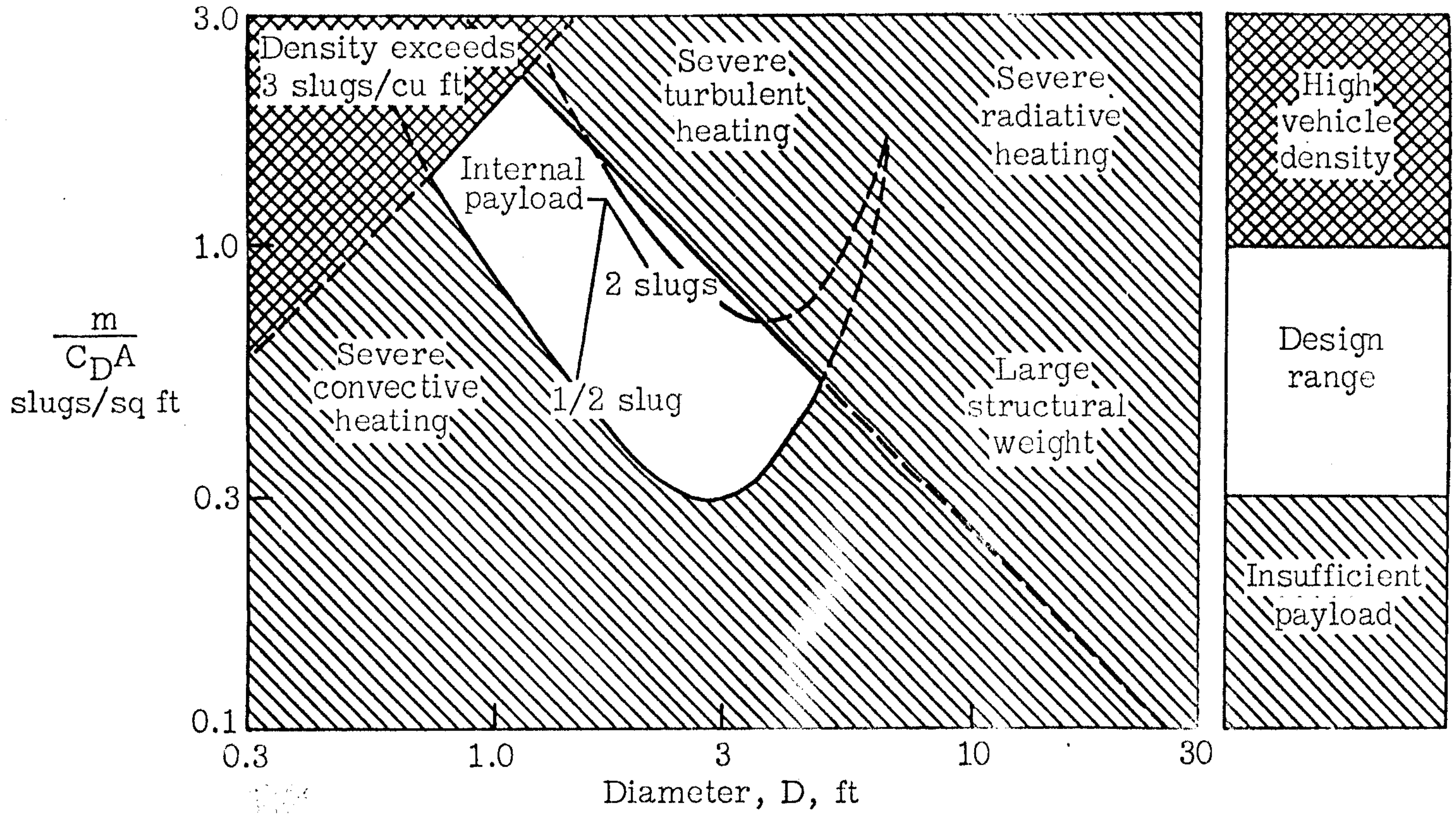

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Figure 7.- Constraints on Venus spherical probe design. (Vertical entry at 40,000 ft/sec.)

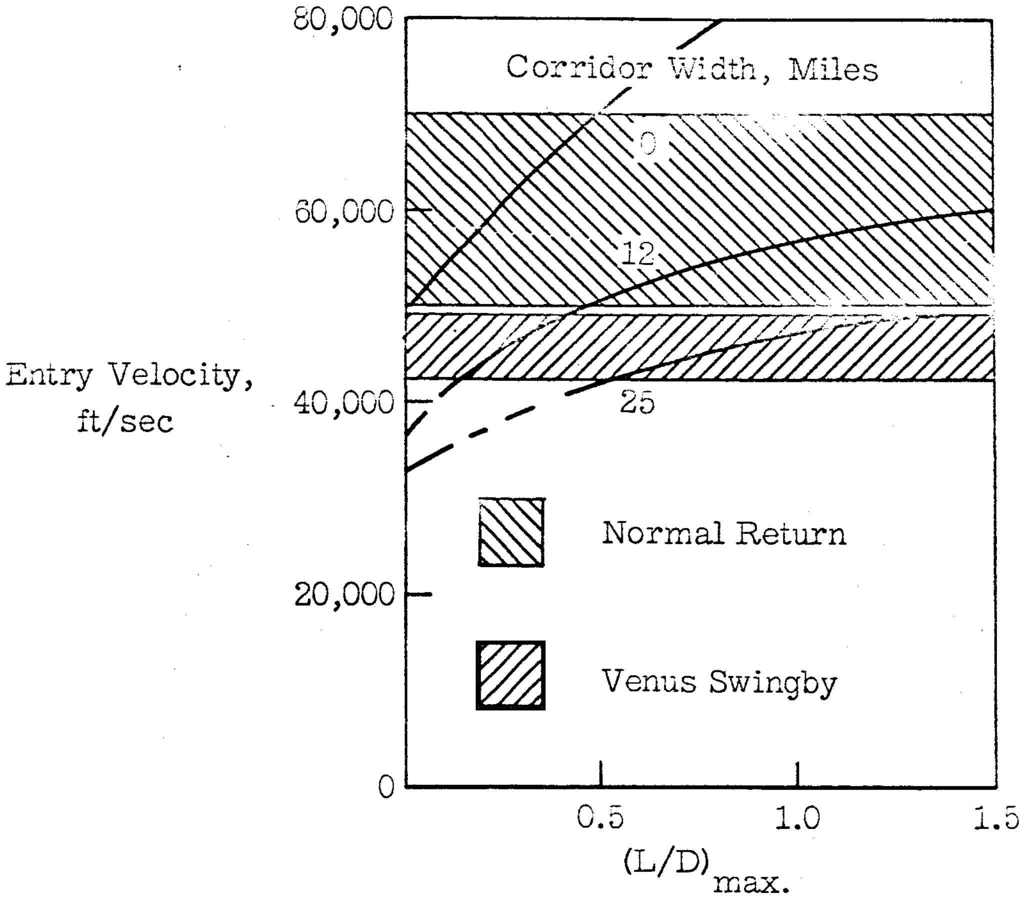

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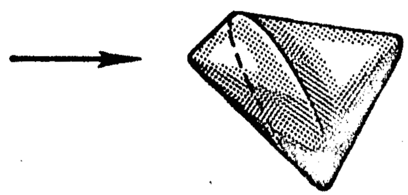
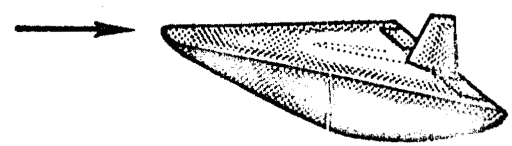
Figure 8.- Mars mission - Earth return corridors.



(a) Conical forebody + elliptical "Apollo."



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(b) Delta planform lifting body.

Figure 9.- Earth entry vehicle concepts.

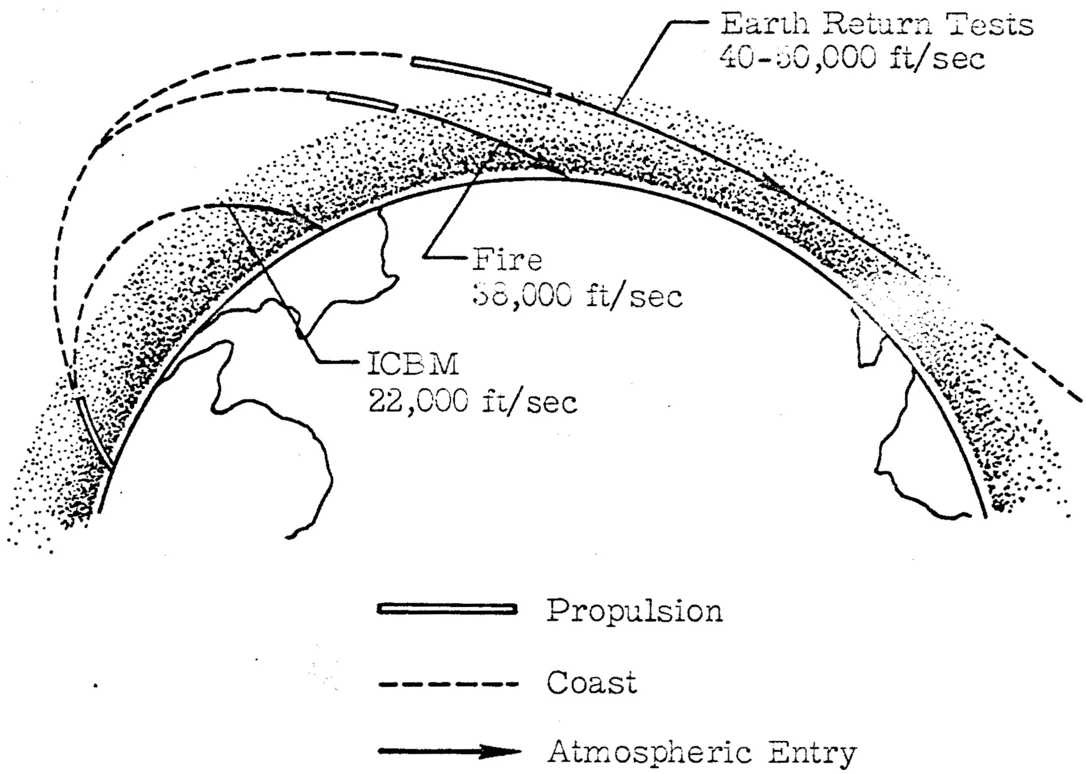

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Figure 10.- Atmospheric entry flight tests.